Structural Sizing of a Horizontal Take-off Launch Vehicle with an Air Collection and Enrichment System

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In support of NASA's Next Generation Launch Technology (NGLT) program, the Andrews Gryphon booster was studied. The Andrews Gryphon concept is a horizontal lift-off, two-stage-to-orbit, reusable launch vehicle that uses an air collection and enrichment system (ACES). The purpose of the ACES is to collect atmospheric oxygen during a subsonic flight loiter phase and cool it to cryogenic temperature, ultimately resulting in a reduced initial take-off weight. To study the performance and size of an air-collection based booster, an initial airplane like shape was established as a baseline and modeled in a vehicle sizing code. The code, SIZER, contains a general series of volume, surface area, and fuel fraction relationships that tie engine and ACES performance with propellant requirements and volumetric constraints in order to establish vehicle closure for the given mission. A key element of system level weight optimization is the use of the SIZER program that provides rapid convergence and a great deal of flexibility for different tank architectures and material suites in order to study their impact on gross lift-off weight. This paper discusses important elements of the sizing code architecture followed by highlights of the baseline booster study.

Nomenclature

h	=	height
D_n	=	area of cross section of surface n
V	=	volume
A,B,C	=	coefficients of second order polynomials
x,y,z	=	axes
p	=	pressure
r	=	radius
t	=	thickness
l	=	length
σ	=	stress
ε	=	strain
α	=	thermal expansion coefficient
E	=	Young's Modulus
T	=	Temperature
m	=	manufacturing tolerance
R	=	LOX to ignition weight ratio
V	=	volume
LOX_1	=	LOX consumed by stage 1 from rocket ignition to shutdown
LOX_2	=	LOX consumed by stage 2 from rocket ignition to shutdown
C_1	=	LOX margin percentage for stage 1 to account for boil-off, flight performance reserve, start up loss,

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fuel bias, etc.
C_2 = LOX margin percentage for stage 2 to account for boil-off, flight performance reserve, start up loss, fuel bias, etc.
r_{JP} = rate of Jet-A fuel consumption during LOX collection
r_{LH} = rate of liquid hydrogen coolant consumption for ACES
JP_{lo} = Jet-A consumption from lift-off to loiter
LH_{bo} = liquid hydrogen boil-off from lift-off to end of LOX collection
C_{bo} = stage 1 LOX boil-off percentage of nominal LOX volume
ρ_{LOX} = LOX density
W = maximum vehicle weight
S = wing reference area
I_{sp} = specific impulse

I. Introduction

NASA has studied advanced launch vehicles under the Next Generation Launch Technology (NGLT) program, including vertical to horizontal take-off, single stage to multistage vehicles, and hypersonic air breathing to multi-fuel engine concepts. The Andrews Space Gryphon conceptual vehicle is an architecture that shows merit for further consideration. The Gryphon is the booster stage that takes off horizontally from a conventional runway with full hydrogen fuel tanks and empty liquid oxygen tanks in order to minimize gross take-off weight (TOGW). An upper stage is mounted tandem atop the Gryphon fuselage. The fuel and oxidizer is crossfed between stages to provide additional TOGW reduction benefit. The Air Collection and Enrichment System (ACES) is on board the booster, and is designed to collect air, cool it to a liquid state, separate the atmospheric constituents, and cryogenically store the oxygen. Four GE90 class turbojet engines are used as compressors in the air collection system, which are fueled by the spent LH_2 coolant. The liquid nitrogen constituent is used to pre-cool the incoming air, and the expanding nitrogen gas is used to provide additional thrust. This occurs during an approximately 2 hour loiter flight phase at an altitude of about 20,000 ft. The LOX collection loiter also provides excellent cross range in nearly all weather conditions. Once sufficient LOX has been collected for a given mission, rockets ignite for a parallel burn of the booster and upper stage to around Mach 5, the upper stage then separates with tanks full, and the booster returns to the launch site, landing horizontally like a conventional aircraft.

To understand the physical size and mass properties of the Gryphon architecture requires an initial geometric shape of the outer mold line (OML) and the allowable wing loading. The Gryphon, illustrated in Fig. 1, is comprised of a basic airplane-like fuselage, swept wings that have vertical control surface winglets, and a lifting canard. Inside the fuselage are the crew compartment and flight deck, landing gear, propellant tanks, and all the associated systems equipment such as telemetry, avionics, thermal control, fuel feed, etc. The ACES is also housed within the fuselage, which includes Heat Exchangers, Rotating Fractional Distillation Unit (RFDU), and four compressor engines. Attached to the aft fuselage bulkhead is a cluster of four derivative Space Shuttle Main Engines (SSME). Mounted to the wings are eight F135 turbojet engines, each fueled by Jet-A and each delivering a maximum thrust of 37,000 lb_f. In order to determine the final Gryphon vehicle size that contains sufficient volume for the amount of fuel required for the mission, an understanding of the fuselage volume, surface areas, and arc lengths of main attachment locations should exist. Then, a precise estimation of the TOGW can be determined, knowing weights per surface area of the vehicle acreage areas, weights per length of attachment zones, and system equipment weights.

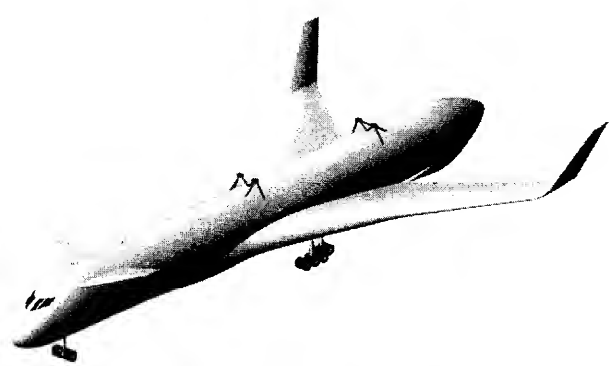

Figure 1. Gryphon Booster.

Weights per surface area, lineal weights of main attachments, and vehicle system equipment weights can be predicted by several means. The top down approach uses a historical database of similar vehicles. In the case of the Gryphon, the weights of various system components such as crew compartment and separation system were estimated with this method. Other vehicle components, such as fuselage sections, wings, etc., are estimated with a

bottom up approach. This approach entails calculating surface areas and arc lengths of the Gryphon vehicle, then applying structural weight factors to those areas and lengths. The weight factors can be determined using a preliminary first order stress analysis.

With either approach, an understanding of the scaling relationships must be known such that sufficient structural weight is predicted in order to maintain strength and stiffness as the vehicle is scaled. For instance, the hoop stress of a pressurized cylindrical tank is proportional to the product of pressure and tank radius and inversely proportional to the tank wall thickness. As the tank radius changes with scale, so must the thickness also change proportionately in order to maintain constant stress. Therefore, the weight per surface area of the tank changes linearly with scale, providing the pressure remains constant.

Once scaling effects on weight and surface area were understood, then parametric relationships of surface area, volume, and thickness were assembled into an excel based spreadsheet. The various areal densities, determined by stress and thermal analyses, were input to the spreadsheet as well. The spreadsheet, called SIZER, was then programmed to predict the final scale of the input vehicle. SIZER places all user input parameters that effect vehicle weight into an organized series of dialogue boxes. SIZER is arranged with numerous options for varying vehicle geometry, engine performance, tank configuration, wing planform, packaging efficiency, and payload.

A. Vehicle Geometry

By representing the OML with a series of second order polynomial curves, SIZER is capable of parametrically scaling any symmetric vehicle shape. The sizing process begins by cutting a vertical section along the keel and fitting eight polynomials to the OML, four windward and four leeward. Next, choosing seven key stations along the keel, the vehicle cross sections are cut. Each half of the seven cross sections is represented by six polynomial curves. Since the vehicle is assumed symmetric along the plane of the keel, the remaining halves of the cross sections are simply a mirror image of the given polynomials. This provides SIZER with a geometric representation of the basic fuselage shape, as shown in Fig. 2, with which to estimate enclosed volumes, external surface areas, and lineal attachment lengths. The means by which SIZER has been coded to make these estimations is explained further.

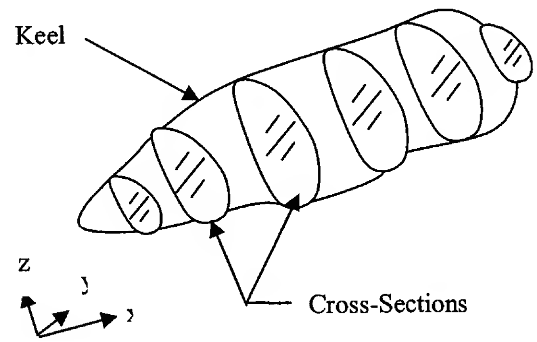

Figure 2. Basic Vehicle Geometry.

With the known geometric shape initially at full scale, SIZER next needs the thicknesses of the vehicle skin, which are illustrated in Fig. 3. These thicknesses account for external tiles of the heat shield, structural skin thickness, any backup rings or stringer thicknesses, internal thermal insulation, any dynamic clearances between the stringers and the propellant tanks, any cryogenic insulation for the tank, and tank skin and stringer thicknesses. Knowing fuselage and tank skin thicknesses allows SIZER to more accurately predict the available internal volume for propellant and equipment. In order to determine volumes, cross section areas are first determined by integrating their representative polynomials. The volume between cross sections of a pyramidal or conic fustrum is then determined by the formula:

$$V = \frac{h}{3}\left(D_1 + \sqrt{D_1 D_2} + D_2\right) \tag{1}$$

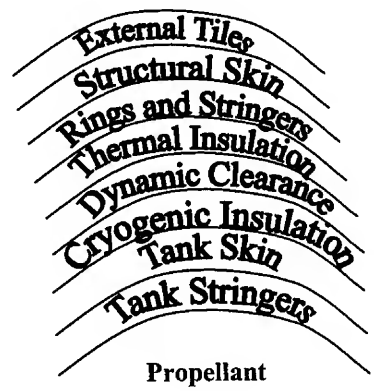

Figure 3. Thicknesses from the Inner to OML.

If the fuselage OML does not have constant taper, i.e. have flat surfaces between the cross sections, then the calculated volume is an approximation. The amount of error is dependent on the amount of curvature in the fuselage. This error can be reduced by decreasing the distance between cross sections, effectively making more cuts throughout the highly curved fuselage.

Other pertinent geometric information is also calculated. The perimeters of the cross sections are calculated from the arc lengths of their polynomials. Fuselage OML surface area between cross sections is approximated by averaging the perimeters of the cross sections multiplied by the average windward and leeward keel lengths between the cross-sections. This provides an exact solution for shapes which, when unrolled, are completely flat. While this

is not exact for other shapes, such as ellipsoids and paraboloids, the error is typically small, especially in the context of conventional vehicle designs where highly curved blunt bodies are not the norm.

B. Vehicle Systems

SIZER includes a library of mass and volume scaling relationships of nearly all the system equipment that launch vehicles require. Various systems that are used for a particular vehicle can be selected while others not used can be deselected. The scaling relationships are generally based on the Air Breathing Launch Vehicle (ABLV) study[1]. Table 1 lists the systems and system abbreviations. Scaling relationships for crew compartment and ejection seats have heritage with other aircraft[2], while stage separation and orbital maneuvering system weights use space shuttle data. If necessary, other systems not found in the library can be added to the vehicle weight summary.

Table 1. System Equipment List.

System Abbreviation	System Description
AVTCS	Air Vehicle Thermal Control System
ECLSS	Environmental Control & Life Support System
EPD&C	Electric Power (incl. generation) Distribution & Control
HYD/ACT	Hydraulics and Actuation
APU	Auxiliary Power Units
RCS	Reaction Control System
FUEL DEL	Fuel Delivery System
OX DEL	Oxidizer Delivery System
VPP&D	Ventilation and Purge Pressurization & Distribution
OMS	Orbital Maneuvering System

C. Propellant Tanks

An important part of sizing vehicles to closure is calculating the propellant tank volumes and surface areas, since these components are typically the largest part of the launch vehicle. SIZER has two basic tank architectures that can be compared in order to investigate their effects on take-off weight. The first tank configuration is the integral tank. This type is defined as the tank whose fluid envelope is structurally tied to the external aeroshell of the vehicle. SIZER assumes that this tank architecture has flat, stiffened bulkheads within the vehicle to contain the fluid. Integral tanks have the best packaging efficiency, but they also have some engineering challenges[3]. For instance, through-the-thickness thermal stresses can be quite severe with cryogenic internal temperatures next to external aerodynamic temperatures. Also, since the bulkheads are flat, they require heavier stiffeners for strength and stiffness requirements than the conventional elliptical end cap designs. The second tank architecture is the conventional tank design termed thermally isolated. This tank is typically nested within the aeroshell by a series of attachment struts. The attachments are designed to allow for thermal disparity between the aeroshell and the tank skin. In addition, there is typically a large dynamic clearance between the tank skin and the aeroshell to account for thermal and structural deformations as well as providing space for inspections and tank removal during ground operations. SIZER represents the conventional tank type as a series of co-joined conic fustrums with elliptical end caps. The eccentricity of the end caps is an input variable. The default value is set to $\sqrt{2}$, which is the ideal theoretical value that gives the lightest end cap structural weight. Other factors have been shown to package propellant more efficiently, in spite of an increase in end cap weight[4].

SIZER automatically calculates the maximum radius to fit a thermally isolated tank within the fuselage, as in Fig. 4.

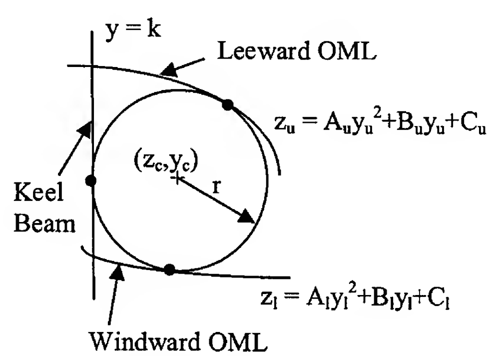

Figure 4. Cross-section of Tank within the Fuselage.

Aeroshell structure, insulation, and tank wall thickness are taken into account. Table 2 lists seven nonlinear equations with seven variables that SIZER simultaneously solves for the location and radius of the largest circle that is tangent to three cross section polynomials. With the radius and center of the circle known, it is a matter of repeating the process by incrementally cross cutting through the fuselage length, determining the tank radius, and calculating the conic volume between cross sections. The tank length is known when the sum of these conic volumes and end cap volumes equals the total required volume.

Table 2. Equations for Determining Circle Tangent to Three Curves

No.	Equation
1	$r=y_c-k$
2	$r^2=(y_c-y_u)^2+(z_c-z_u)^2$
3	$r^2=(y_c-y_l)^2+(z_c-z_l)^2$
4	$(y_c-y_l)/(z_c-z_l)=-(2A_l y_l+B_l)$
5	$(y_c-y_u)/(z_c-z_u)=-(2A_u y_u+B_u)$
6	$z_l=A_l y_l^2+B_l y_l+C_l$
7	$z_u=A_u y_u^2+B_u y_u+C_u$

The required tank volume is based on the propellant density and propellant mass fraction. In addition, SIZER also takes into account other factors that effect tank volume, such as pressure and thermal deformations, manufacturing tolerances, and ullage. Cylindrical tank pressure deformations in the hoop and longitudinal directions are as follows:

$$\sigma_{hoop} = \frac{pr}{t} \tag{2}$$

$$\sigma_{long} = \frac{pr}{2t} \tag{3}$$

Stress is generally

$\sigma = E\varepsilon$, where strain is (4)

$\varepsilon_{hoop} = \dfrac{\Delta r}{r}$ and (5)

$$\varepsilon_{long} = \frac{\Delta l}{l}. \tag{6}$$

Substituting Eqs. (2) and (5) into Eq. (4) and solving for the change in radius gives

$$\Delta r = \frac{pr^2}{tE_{hoop}} \tag{7}$$

Likewise, substituting Eqs. (3) and (6) into (4) and solving for the change in length gives

$$\Delta l = \frac{prl}{2tE_{long}} \tag{8}$$

The change in volume of a cylinder due to pressure can be approximated as

$$\Delta V \approx \Delta l\left(\pi r^2\right) + 2\pi r l \Delta r \tag{9}$$

Substituting (7) and (8) into (9) gives

$$\Delta V \approx \pi\left(\frac{pr^3 l}{2tE_{long}} + \frac{2pr^3 l}{tE_{hoop}}\right) \tag{10}$$

$$\frac{\Delta V}{V} \approx \frac{pr}{t}\left(\frac{1}{2E_{long}} + \frac{2}{E_{hoop}}\right) \tag{11}$$

Following a similar logic, the thermal volumetric change is determined as

$$\frac{\Delta V}{V} \approx 2\alpha_{hoop}\Delta T + \alpha_{long}\Delta T \tag{12}$$

In order to account for manufacturing tolerances over three axes, length, width, and height, the normalized change in volume is approximately

$$\frac{\Delta V}{V} \approx 3m \tag{13}$$

The machining tolerance m is typically on the order of 0.001 in. per in. of part size. Using typical values for conventional materials and machining tolerances for the tanks, the combined expected change in volume for pressure, temperature and manufacturing is about a 1% loss in the nominal volume. On top of this loss, propellant ullage, usually on the order of 1%, must also be accounted. Therefore, a tank should be manufactured to have a volume about 2% larger than the nominal size to ensure sufficient volume for propellant. SIZER calculates the precise volumetric variance from the material properties, ullage, and manufacturing tolerance requirements.

D. Support Structure

Bulkheads or rings are placed within the fuselage to offer backup structure to the aeroshell. A key input to SIZER is the specified distance between bulkheads, which is automatically maintained regardless of vehicle scale. Likewise, the space between longerons can be held constant. In this way, the aeroshell panel size between stiffeners remains constant so that its allowable buckling strength remains constant even as the vehicle scales. If the scale of the vehicle increases to closure, more bulkheads, and therefore more structural weight is added to the backup structure. Since only the radius of curvature and not panel dimensions changes with scale, the aeroshell weight per surface area remains unchanged, especially if the panel weight was assessed using flat plate theory.

E. Aerodynamic Surfaces

Vehicles with aerodynamic surfaces such as wings, winglets, canards and tail are sized using classical aircraft design methods[5]. Wings are sized using a fixed wing loading value, *W/S*. The reference area S encompasses the leading and trailing edges of the wing from the wingtip extending to the fuselage centerline, but not including strakes. The basic wing size geometry is estimated by the taper ratio, aspect ratio, and root thickness ratio. Horizontal and vertical tails are sized by the volume coefficient method. Lifting type canards are sized with the wing area split method, as in the case of the Gryphon.

Wing weight and wing tank volume are a function of the actual wing planform area that extends from the fuselage surface to the wing tips, including strakes. Currently, SIZER does not treat the weights of flaps, slats, speed brakes and other hydraulically actuated controls separately, but includes that weight as part of the whole wing. Future releases will address these control surfaces separately from the entire wing. To more accurately predict available volume for fuel, slats, flaps, and landing gear envelopes are accounted in the wing tank volume calculation.

Since aerodynamic surface areas are driven by vehicle weight, they do not scale the same as the fuselage. Therefore, vehicle lift and drag characteristics tend to change during the iterative closure process. This must be kept in mind between trajectory analysis and sizing analysis. To aid in the closure process, future releases of SIZER will estimate the coefficients of lift and drag as the vehicle rescales.

F. Landing Gear

Conventional jet aircraft landing gear are hydraulically actuated wheels in the nose and wing locations. SIZER calculates the required wheel diameter based upon the predicted wheel loads, number of wheels, location of the gear, and load rating of the wheels. The wheel load rating, wheel weight per inch of diameter, and number of wheels are input variables. SIZER determines the TOGW, calculates the vehicle center of gravity, predicts landing loads in the nose gear and main gear, and then sizes the wheels accordingly. Based on the diameter of the nose wheel, SIZER determines how close to the nose tip it can be placed using the same methodology of finding the location of a circle tangent to the fuselage OML as the tank diameter calculation. Nose landing gear envelope is accounted when calculating the available volume within the fuselage. If parachutes or airbags are used in the design, their envelopes and weights can be estimated simply by adjusting the wheel number, the weight per diameter, and load and weight rating values.

Main landing gear is outboard of the fuselage located in the deep section of the wing box. Since it is outside the environmentally controlled fuselage, SIZER does not currently check the fit of the gear within the wing. Like the nose gear, main gear weights are determined by specifying the number of wheels, load, and weight ratings. It must be kept in mind, though, that SIZER does not limit the maximum wheel diameter but leaves it up to the engineer to ensure his design is capable of being manufactured and assembled.

G. Air Collection and Enrichment System

The ACES is unusual because in that LOX mass is added to the aircraft during the mission. In order to scale the vehicle with ACES aboard requires the software to determine the necessary LOX tank volume. It is assumed that ACES collects LOX and consumes hydrogen coolant at a fixed, average rate. Likewise, jet fuel consumption can be averaged throughout the loiter duration. Therefore, determining the LOX tank volume then becomes a matter of knowing the oxidizer mass to ignition mass ratio, taking into account margins for boil-off, start up losses, etc. This ratio is found from trajectory analysis and is defined as follows:

$$R = \frac{LOX_1}{Ignition\,Wt} \tag{14}$$

$$\begin{aligned} Ignition\,Wt &= TOGW + LOX_1 + LOX_2 + C_1 LOX_1 + C_2 LOX_2 - \\ &(r_{JP} + r_{LH})(LOX_1 + LOX_2 + C_1 LOX_1 + C_2 LOX_2) - JP_{lo} - LH_{bo} - C_{bo} LOX_1 \end{aligned} \tag{15}$$

Solving for LOX_1 and factoring in its density gives the booster tank volume:

$$V_1 = \frac{R(TOGW + (1 + C_2)(1 - r_{JP} - r_{LH})LOX_2 - JP_{lo} - LH_{bo})}{(1 + R(C_{bo} + (r_{JP} + r_{LH})(1 + C_1) - (1 + C_1)))\rho_{LOX}} \tag{16}$$

This formula applies to a two stage to orbit vehicle, taking off with LOX tanks initially empty.

II. Closure

Having determined structure, equipment, tank sizes, and their scaling laws, it is now possible to predict the closure weight and size of a vehicle for a given trajectory. A vehicle is closed when it has sufficient volume to meet the propellant fraction requirements with sufficient structure to withstand the mission flight loads. Closure is an iterative process between aerodynamics, trajectory, sizing and structural analysis. As a vehicle size approaches closure, the wing reference area scales as a function of weight while the fuselage scales with volume requirements. As a result, the vehicle lift and drag characteristics change and will have to be reconsidered during the closure iteration if sufficient change has been made to the outer mold line. Convergence to closure is typically rapid, usually within 2 or 3 iterations.

SIZER makes all the geometric, mass, and center of gravity calculations to predict the weight of the entire vehicle at any given scale. In addition, basic fit checks are performed to ensure sufficient space is available to contain the required payload size and volume, as well as sufficient area on the aft bulkhead to attach the required

number of rocket engines. SIZER either rubberizes the engines or discretely adds fixed engines as needed in order to maintain the minimum required thrust-to-weight ratio for the vehicle.

Specific impulse (I_{sp}) and the change in velocity of the vehicle are key inputs to SIZER used to determine closure. When available, I_{sp} is determined by Optimal Trajectories by Implicit Simulation (OTIS)[6] for the given engine performance. Otherwise I_{sp} is estimated with engine performance maps and range data. Using I_{sp} and the change in velocity from the trajectory calculations in OTIS, the rocket equation provides the mass ratio of propellant to the lift–off mass.

Shown in Fig. 5 is a flow chart of the closure process within SIZER. Essentially, the size of the vehicle is determined by the payload mass and size requirements, structural efficiency of the vehicle itself, the propulsive performance of the engines that lift the vehicle through its flight, and the aerodynamic characteristics of that vehicle. SIZER closes a vehicle by first calculating the TOGW and the propellant weight required to fly the mission and then by comparing the tank volume to the propellant volume. If the propellant volume required is not equal to the tank volume available, the vehicle is scaled accordingly until the equality is met.

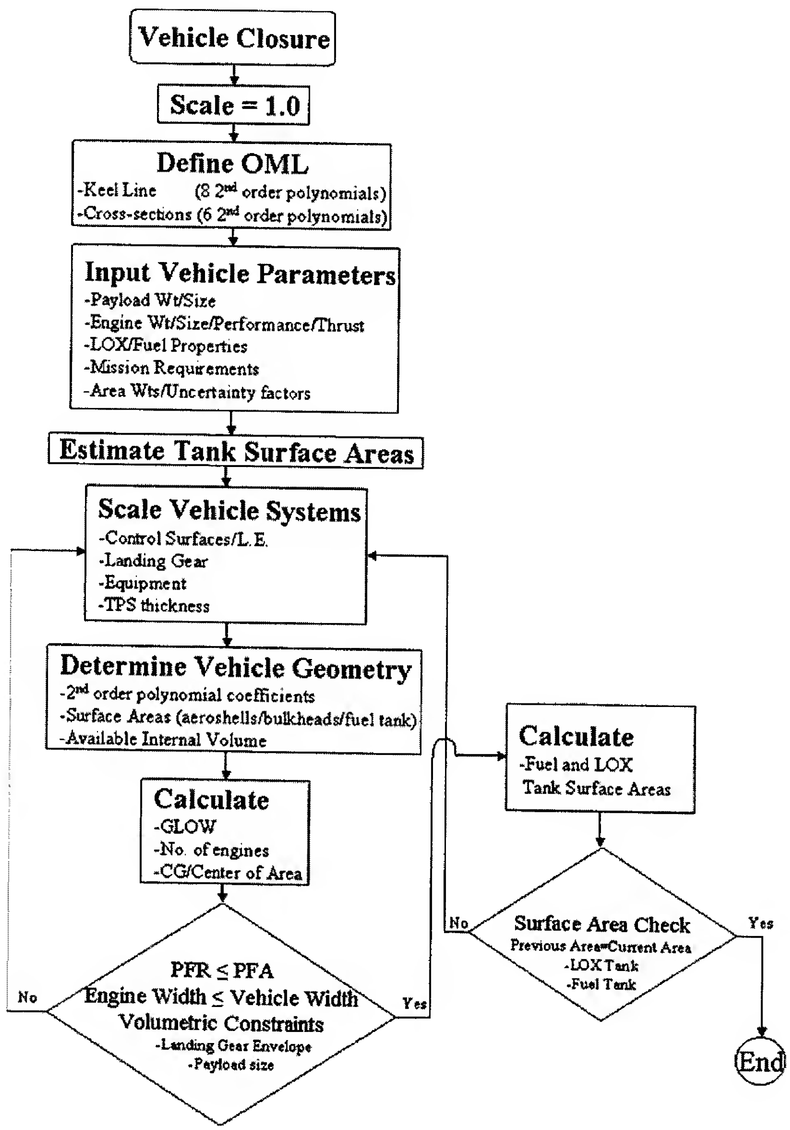

Figure 5. SIZER Closure Logic.

III. Gryphon Vehicle Results

Starting with an OML definition provided by Andrews Space, the vehicle OML was entered into SIZER. Next, the numerous variables that affect the size of the vehicle were input in order to determine the closed TOGW and maximum payload delivered to orbit. Gryphon uses two thermally isolated fuel tanks where the forward tank contains 75% of the fuel. A single oxidizer tank is nested between the fuel tanks and is located near the wing. The ACES is located near the vehicle nose. Together these features provide a stable center of gravity for all flight conditions. Fig. 6 shows the pertinent input variables used to synthesize the Gryphon.

The final take-off weight of the combined stages is 1.255 e6 lbs with a wingspan of 191 ft and a nosetip to aft bulkhead vehicle length of 263 ft. The body diameter is slightly greater than 30 ft. The tank diameters are about 27.5 ft. Figure 7 is generated by SIZER and shows a side view of the closed vehicle fuselage. Nose gear, main bulkhead, and propellant tank locations are depicted in the figure. Elliptical end caps on the propellant tanks are currently represented with vertical lines. Future releases of SIZER will draw the actual shape of the end caps. Table 3 gives a summary of the SIZER output results for this vehicle, including first and second stage fuel usage.

The size of the Gryphon has heritage with several large aircraft that are in service. The Boeing 747-400ER/ERF has a wingspan of 211 ft, a length of 231 ft and a maximum take-off weight of 910,000 lbs. Another large aircraft is the Russian Antonov 225, deployed around 1989. The An-225 was built to carry the Buran Space Shuttle atop its fuselage much like the Gryphon is designed to carry its upper stage. The An-225 has a maximum TOGW of 1.322 e6 lbs, a wing span of 290 ft and a length of 276 ft. So, while the Gryphon is an extremely large aircraft, it would not be the largest ever built. However, if it should be built, it would be the largest aircraft to fly supersonically.

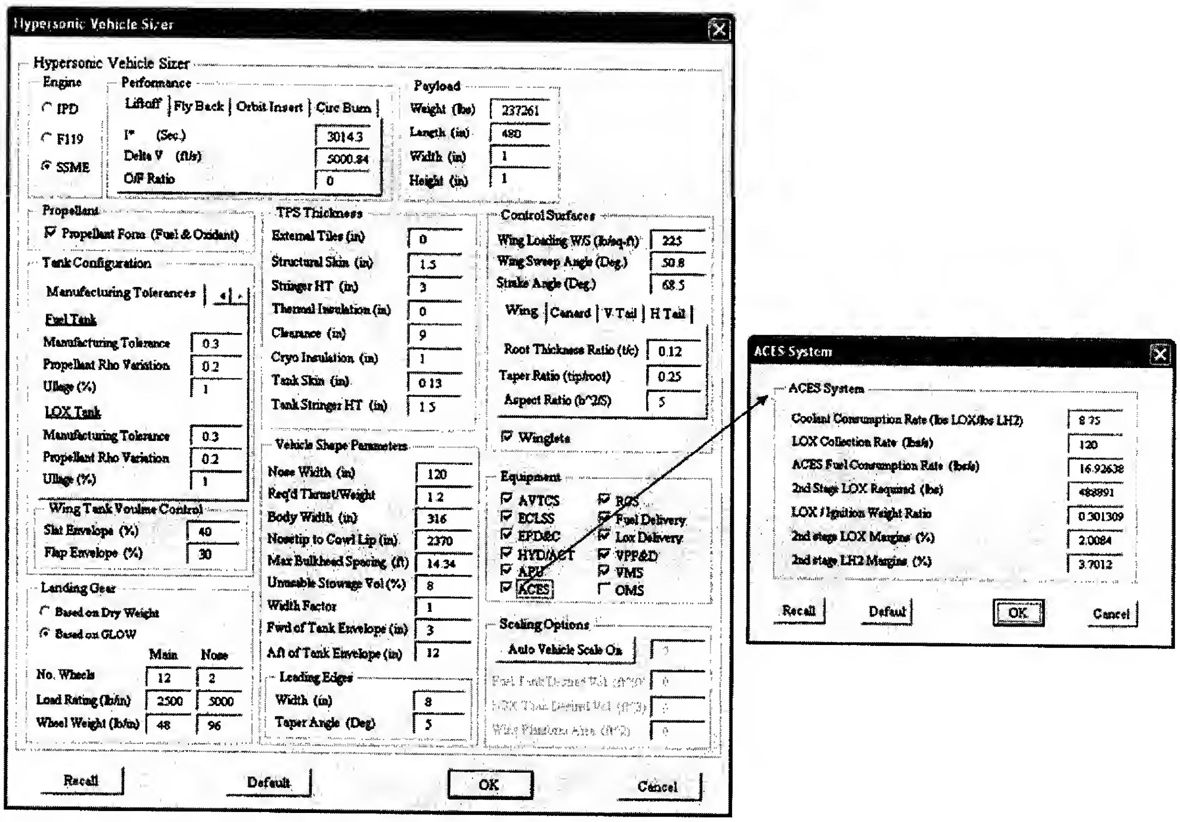

Figure 6. SIZER Dialogue Boxes Showing Inputs for Gryphon.

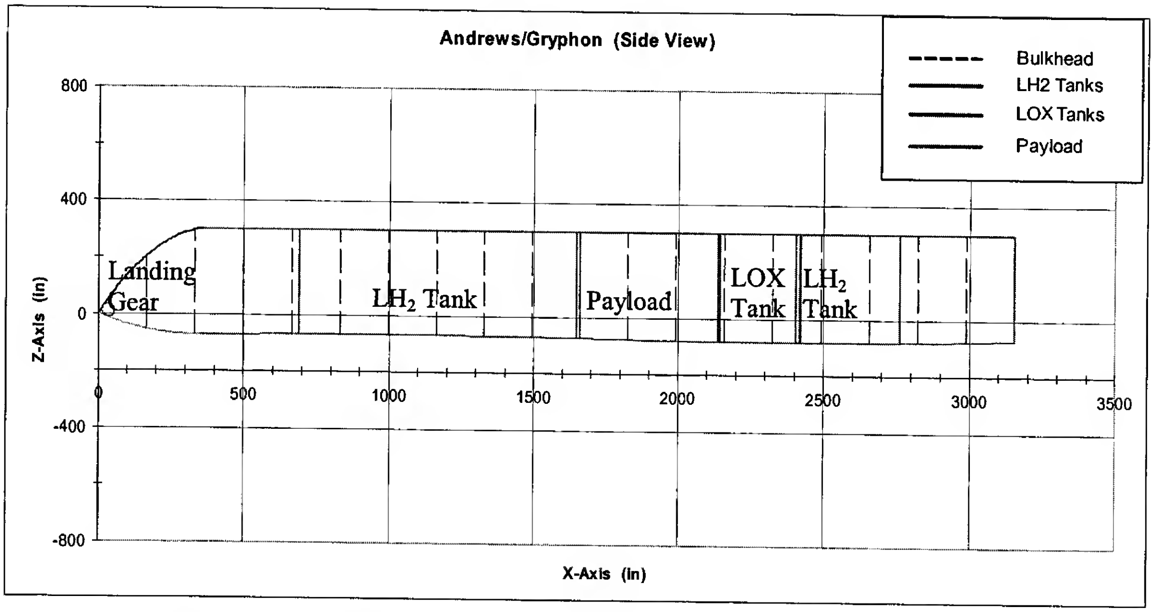

Figure 7. Side View of the Fuselage for the Closed Gryphon Launch Vehicle.

Table 3. SIZER Gryphon Weights Summary.

Version	1.3		
Release Date	3/8/2004		
Date	3/26/2004		
Scale Factor	1.150		
GLOW	1,255,172 lbs		
Vehicle Length	263 ft		
Ignition Weight	2,055,697 lbs		
Wing Area	8763.6 ft^2	7291 wing	canard 1473
Engines	4 SSME		
Material	Gr BMI		
Fuel	NBP		
LOX	NBP		
Fuel Tank Type	Isolated Tank		
LOX Tank Type	Isolated Tank		

	Fuel Schedule lbs	**Vehicle Weights** lbs
Take-off		1,255,172.20
LOX Collection (for 1st and 2nd stage engines + reserves)	1,132,211.67	
LOX Margin + Orbit	42,099.06	
Jet-A Usage from Take-off to End of LOX Collection	180,723.66	
Jet-A Usage from Take-off to Start of LOX Collection	21,021.62	
LH2 Usage for ACES	129,395.62	
Startup losses		
LH2 Boil-off (Total accounted to first stage)	8,486.30	
LOX Boil-off (Total accounted to first stage)	13,081.10	
at End of LOX Collection		2,055,697.19
JP Usage for Ascent Segment	9,092.35	
LH2 Usage for 1st Stage	103,233.34	
LOX Usage for 1st Stage	619,400.06	
at Staging		1,323,971.44
Tail Cones		5,931.53
Booster Flight Back Weight + Tail Cones		597,819.44
Booster Flight Back Weight - Tail Cones		591,887.91
2nd Stage Weight at Staging		726,152.00
Propellant consumed by 2nd stage	559,143.01	
LH2 Usage for 2nd Stage	79,877.57	
LOX Usage for 2nd Stage	479,265.43	
Discarded payload fairing weight		7,618.17
2nd Stage Weight at orbit		159,390.83
2nd stage dry weight + payload		146,808.83
2nd stage fuel Margin (includes deorbit burn)	2,956.43	
2nd stage lox margin + orbit/deorbit	9,625.57	

Acknowledgments

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